NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE 3261

A METHOD FOR EVALUATING THE EFFECTS OF DRAG AND INLET PRESSURE RECOVERY ON PROPULSION-SYSTEM PERFORMANCE

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Washington
August 1954

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SUMMARY

A method of evaluating the ratio of aircraft thrust minus drag to ideal thrust for any inlet pressure recovery is presented for airplanes with air-breathing propulsion systems. The quantities required for this evaluation include flight Mach number and altitude, engine total-pressure ratio and total-temperature ratio, engine air-flow requirements, and airplane drag. Variation of the ratio of incremental changes in inlet pressure recovery to incremental changes in drag coefficient required to maintain a given level of airplane or propulsion-system thrust minus drag is presented. Three types of typical air-breathing engine having equal air-flow capacities are compared at given operating points. Application of the method to the determination of inlet mass-flow ratio for maximum thrust minus drag is also included.

INTRODUCTION

The air-induction system of an aircraft with an air-breathing engine should supply the prescribed air flow to the engine at high pressure recovery with as little external drag as possible. It is often possible to increase the internal thrust of the propulsion system through inlet design modifications that increase the operating pressure recovery of the inlet. If such an increase in pressure recovery is obtained without increasing the airplane drag coefficient, an increase in airplane thrust minus drag will result. If, however, an increase in airplane external drag coefficient accompanies the increase in inlet pressure recovery, the resultant effect on the airplane thrust minus drag cannot be determined without further evaluation of the propulsion-system parameters.

A method of evaluation of air induction systems combined with arbitrary jet engines from considerations of induction-system air-handling qualities and engine component characteristics is presented in reference 1. Reference 2 presents a simplified method for comparing the performance of supersonic ram-jet diffusers. The purpose of the present report

is to present a method for evaluating the effects of inlet pressure recovery and drag on propulsion-system-thrust-minus-drag performance from considerations of engine over-all "pumping" characteristics.

The analysis presented herein applies to a complete airplane or its propulsion-system components where selection of, or design modifications to, the inlet are being considered. Inlet design considerations such as incorporation of boundary-layer removal systems on scoop-type inlets, inlet location on the aircraft, and choice of cowl fairing and lip angle will determine the level of pressure recovery at which the inlet will operate together with the associated airplane external drag coefficient. If the various levels of pressure recovery and drag coefficient for these design considerations are known, the equations and curves presented herein facilitate the choice of inlet for maximum thrust minus drag. Other factors that may be affected by inlet design modifications such as changes in airplane and component weights or changes in engine fuel consumption are not considered.

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SYMBOLS

The following symbols are used in this report:

A	flow area
A_c	compressor-inlet flow area of turbojet engine
A^*	flow area at $M = 1.0$
C_D	drag coefficient, D/q_0S
ΔC_D	incremental drag coefficient
C_F	internal thrust coefficient, F/q_0A_0
D	drag
F	internal thrust of engine and inlet combination
f/a	fuel-air ratio
k	slope of internal thrust ratio - pressure recovery curve
M	Mach number
m	mass-flow rate
m/m_0	inlet mass flow ratio, unity when free-stream tube as defined by cowl lip enters inlet

N/N^*	ratio of actual to rated mechanical engine speed
P	total pressure
$\Delta(P_2/P_0)$	inlet incremental pressure recovery
p	static pressure
q_0	free-stream dynamic pressure, $\frac{\gamma_0}{2} p_0 M_0^2$
R	gas constant
S	airplane drag coefficient reference area
T	total temperature, $^{\circ}R$
t	static temperature, $^{\circ}R$
v	flow velocity
w_a	weight flow of air passing through engine
γ	ratio of specific heats
δ	$P/2116$
θ	$T/519$
μ	$1 + f/a$

Subscripts:

0	free-stream
1	inlet entrance
2	diffuser exit
e	nozzle exit
i	ideal inlet operation, $P_2/P_0 = 1.0$

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ANALYSIS

One type of propulsion-system effectiveness can be determined from the difference between internal thrust and the associated configuration or component drag. If a fully expanded exhaust nozzle[1] with no total-pressure loss is assumed for simplicity, the propulsion system internal thrust is related to the efficiency of the air induction system. With the station designations of figure 1, a thrust parameter for air-breathing engines $\frac{C_F + 2}{\sqrt{T_e/T_2}}$ is derived in appendix A. This parameter is a function only of free-stream Mach number and propulsion-system over-all total-pressure ratio. A linear approximation to equation (B2) for the internal thrust ratio F/F_i was made in appendix B for a limited range of inlet total-pressure recovery (fig. 2). This range is probably applicable to most inlets designed for the Mach number range considered herein (up to free-stream Mach number M_0 of 3.0). For higher Mach numbers, other approximations to equation (B2) may be necessary.

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A set of curves for the evaluation of ideal internal thrust coefficient $C_{F,i}$ (based on free-stream tube area for inlet pressure recovery P_2/P_0 of 1.0) and slope of the thrust ratio - pressure recovery curve k of figure 2 obtained from the equations derived in the appendixes is presented in figure 3. The curves in quadrant I (fig. 3) may be entered with $(P_e/P_0)_i = P_e/P_2$ and M_0 to obtain $\frac{C_{F,i} + 2}{\sqrt{T_e/T_2}}$. Reading to the right in quadrant II with T_e/T_2 yields $C_{F,i}$. Entering quadrant III with $(P_e/P_0)_i$ and M_0 gives the ratio $\left(\frac{C_F + 2}{C_{F,i} + 2}\right)_{P_2/P_0=0.667}$, which, together with $C_{F,i}$, determines k in quadrant IV.

Generalized engine pumping characteristics for a typical turbojet engine are presented in figure 4. From curves such as these, engine total-pressure ratio P_e/P_2 and total-temperature ratio T_e/T_2 can be determined for any corrected-engine-speed ratio $\frac{N/\sqrt{\theta}}{N^*}$ and used with figure 3 to obtain $C_{F,i}$ and k, as discussed previously. Knowing k, the variation of internal thrust ratio with inlet pressure recovery

[1]The assumption of a fully expanded exhaust nozzle results in values of propulsion system thrust coefficient substantially the same as those obtained with a sonic nozzle for flight Mach numbers up to about unity. As flight Mach numbers are increased into the supersonic region, however, considerably higher values of thrust coefficient are obtained with the fully expanded exhaust nozzle.

P_2/P_0 is defined. If the generalized engine characteristics are unavailable, the engine total-pressure and total-temperature ratios for a given set of engine operating conditions can usually be obtained from the engine manufacturers' performance charts.

In order to evaluate the thrust minus drag of an aircraft, the configuration drag must also be known. A convenient expression for the evaluation of thrust minus drag is $\frac{F - D}{F_i}$. The quantity F/F_i has been
3328 discussed in the preceding section, so that only D/F_i, which is defined as follows remains to be considered:

$$\frac{D}{F_i} = \frac{C_D S}{C_{F,i} A_{0,i}}$$

Airplane drag coefficient C_D will, of course, be influenced by such factors as the aerodynamic shape of the airplane, the flight Mach number and altitude, and the inlet mass-flow ratio. The inlet size is determined from inlet-engine matching criteria (ref. 3). Once the inlet size has been selected, the drag associated with the inlet will be determined by the air-flow requirements of the engine. Typical turbojet-engine air-flow requirements are shown in figure 4(b) as a function of corrected-engine-speed ratio. In order to evaluate an airplane drag coefficient at a given flight Mach number and altitude, the variation of drag coefficient with engine air-flow requirements must be known.

With engine air-flow requirements obtained from figure 4(b), the ratio of ideal free-stream tube area to compressor-inlet flow area $A_{0,i}/A_c$ can be determined from the curves of figure 5 based on continuity relations:

$$A_{0,i}/A_c = \frac{0.02022 \, \frac{w_a\sqrt{\theta}}{\delta_c A_c}}{(A^*/A)_0}$$

Knowing the compressor inlet flow area A_c, it is then possible to solve for $A_{0,i}$.

The evaluation of $C_{F,i}$ has been discussed in the preceding section and S is a known quantity for a given aircraft. Thus all quantities necessary for the determination of D/F_i have been discussed.

By definition, $C_{F,i}A_{0,i} = F_i/q_0$. Consequently, the term $C_{F,i}A_{0,i}$ may also be evaluated by dividing the engine ideal thrust by q_0, providing such information is more readily available than that presented in the foregoing paragraphs.

The quantity $\frac{F - D}{F_i}$ for a given airplane-engine combination has been shown to be a function of inlet pressure recovery and airplane drag. For an increase in inlet pressure recovery and/or a decrease in airplane drag, $\frac{F - D}{F_i}$ obviously increases. If the inlet pressure recovery and airplane drag simultaneously increase or decrease, their combined effect on $\frac{F - D}{F_i}$ is not immediately apparent and the individual terms must be evaluated to determine the trend of this quantity. It can be pointed out, however, that if $\frac{F - D}{F_i}$ remains unchanged, an increase in inlet pressure recovery and drag coefficient would increase the propulsion system fuel consumption because of the increase in size of the free-stream tube of air passing through the inlet at a given engine operating point (appendix B).

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It may be of interest to obtain the increment in inlet pressure recovery required to overcome a certain increment in airplane drag coefficient associated with an airplane design modification. For this condition, the quantity $\frac{F - D}{F_i}$ remains constant:

$$\frac{F - D}{F_i} = k\frac{P_2}{P_0} + (1 - k) - \frac{C_D S}{C_{F,i}A_{0,i}} = \text{constant} \tag{1}$$

or

$$\Delta\frac{(F - D)}{F_i} = k\,\Delta\left(\frac{P_2}{P_0}\right) - \frac{\Delta C_D S}{C_{F,i}A_{0,i}} = 0 \tag{2}$$

then

$$k\,\Delta\left(\frac{P_2}{P_0}\right) = \frac{\Delta C_D S}{C_{F,i}A_{0,i}}$$

and

$$\frac{\Delta(P_2/P_0)}{\Delta C_D} = \frac{S}{kC_{F,i}A_{0,i}} \tag{3}$$

For a given airplane then, where S remains fixed, the quantity $\frac{\Delta(P_2/P_0)}{\Delta C_D}$ is a function of the factors k, $C_{F,i}$, and $A_{0,i}$ as determined

in the preceding sections. Thus, if the engine pumping characteristics (fig. 4) and compressor-inlet area A_c are known, $\frac{\Delta(P_2/P_0)}{\Delta C_D}$ can be evaluated for any free-stream Mach number, altitude, and engine operating point.

As will be discussed later, equation (3) can also be applied to the
3328 evaluation of inlet mass-flow ratio for maximum $\frac{F - D}{F_i}$.

Illustrative Example

An illustration of the evaluation of $\frac{\Delta(P_2/P_0)}{\Delta C_D}$ can be obtained by considering a turbojet engine (no afterburner) with pumping characteristics as shown in figure 4. For this example, the following assumptions are made:

Free-stream Mach number, M_0	2.0
Altitude (isothermal region of atmosphere), ft . . .	35,000 and above
N/N^*	1.0
t, $^{\circ}$R (ref. 4)	392°
t/T (ref. 4)	0.5556

From this information, $T_0 = T_2 = 706^{\circ}$ R and $\sqrt{\theta} = 1.166$. Corrected engine-speed ratio

$$\frac{N/\sqrt{\theta}}{N^*} = \frac{1.0}{1.166} = 0.857$$

From figure 4,

$$P_e/P_2 = 1.41$$

and

$$T_e/T_2 = 1.97$$

Entering quadrant I in figure 3 with P_e/P_2 or $(P_e/P_0)_i$ of 1.41 and M_0 of 2.0 and reading across into quadrant II with T_e/T_2 of 1.97, yields a value for $C_{F,i}$ of 1.15. Quadrant III may then be entered with the same values of P_e/P_2 and M_0 (1.41 and 2.0, respectively) to obtain the ratio $\left(\frac{C_F + 2}{C_{F,i} + 2}\right)_{P_2/P_0=0.667}$. With this ratio and $C_{F,i}$

already determined, k may be evaluated in quadrant IV. Dashed lines with arrows have been drawn on figure 3 to indicate the procedure to be followed in reading the curves. For this example, a value of k of 1.37 is obtained.

From equation (3),

$$\frac{\Delta(P_2/P_0)}{\Delta C_D} = \frac{S}{kC_{F,i}A_{0,i}} = \frac{S/A_{0,i}}{1.37\times1.15}$$

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and

$$\frac{\Delta(P_2/P_0)}{\Delta C_D} = 0.634\ \frac{S}{A_{0,i}}$$

Variation of $\frac{\Delta(P_2/P_0)}{\Delta C_D}\frac{A_{0,i}}{S}$ with free-stream Mach number M_0 is presented in figure 6. Curves for the turbojet engine with afterburner and for a ram-jet engine have also been included. A nozzle-exit total temperature T_e of 3500° R was assumed for these two engines. The comparison for all these engines is based on equal air-flow handling capacities.

Values of $\frac{\Delta(P_2/P_0)}{\Delta C_D}\frac{A_{0,i}}{S}$ on the curves of figure 6 are a function only of the engine total-pressure and total-temperature ratios for a given flight Mach number and altitude. For the operating conditions chosen, the value of $\frac{\Delta(P_2/P_0)}{\Delta C_D}\frac{A_{0,i}}{S}$ at a free-stream Mach number of 2.0 for the turbojet engine (no afterburner) is approximately twice that for the ram-jet engine or turbojet engine with afterburner. This comparison would change, of course, for a change in the operating point of any of the engines. For a turbojet-powered airplane using the wing area as a reference, a typical value of $S/A_{0,i}$ might be 25. The quantity $\frac{\Delta(P_2/P_0)}{\Delta C_D}$ for this value of $S/A_{0,i}$ at a free-stream Mach number of 2.0, is 15.8 (fig. 6). To maintain a constant value of $\frac{F - D}{F_i}$ then, an increase of 0.001 in drag coefficient would require an increase of 0.0158 in inlet pressure recovery. If $\frac{\Delta(P_2/P_0)}{\Delta C_D}$ for the actual airplane is greater than 15.8, the pressure-recovery increase will outweigh the increase in drag coefficient and $\frac{F - D}{F_i}$ will increase. The converse of this statement is also true.

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Inlet mass-flow ratio for optimum thrust minus drag of the inlet-engine combination may also be obtained from equation (1). Maximum thrust minus drag occurs when $\dfrac{d\left(\dfrac{F - D}{F_i}\right)}{d(m/m_0)} = 0$. This relation leads directly to equation (3) in differential form and, if expressed in terms of inlet mass-flow ratio,

$$\frac{d(P_2/P_0)}{d(m/m_0)} = \frac{S}{kC_{F,i}A_{0,i}} \frac{dC_D}{d(m/m_0)} \tag{4}$$

From the assumptions of the foregoing example, $\dfrac{S}{kC_{F,i}A_{0,i}} = 15.8$. A realistic value of $\dfrac{dC_D}{d(m/m_0)}$ (at $M_0 = 2.0$) would be on the order of -0.5 based on cowl-lip area (see ref. 5, e.g.). If the reference area is converted to wing area, $\dfrac{dC_D}{d(m/m_0)}$ becomes -0.0167 for a ratio of wing area to cowl-lip area of 30. Thus $\dfrac{d(P_2/P_0)}{d(m/m_0)} = -0.264$ for the preceding conditions. Consequently, the point on the inlet pressure recovery - mass-flow-ratio curve where the slope is equal to -0.264 defines the inlet mass-flow ratio for which a maximum value of $\dfrac{F - D}{F_i}$ is obtained. For the inlet of figure 7, this mass-flow ratio is approximately 0.82 as determined from the point of tangency of the dashed line (slope = -0.264) and the pressure-recovery curve. The optimum mass-flow ratio obtained in this manner represents a more rapid process than that of calculating the thrust minus drag for a range of mass-flow ratios to determine the maximum value. Variation of inlet drag coefficient (based on inlet capture area) with mass-flow ratio is also included in figure 7. This variation is not necessarily linear as shown for the typical inlet chosen, but a linear approximation to a nonlinear drag coefficient curve can probably be made for the range of inlet mass-flow ratios concerned. It is interesting to note that the mass-flow ratio for maximum $\dfrac{F - D}{F_i}$ occurs in the region between critical (minimum drag) and peak pressure recovery.

CONCLUDING REMARKS

The derivations and the curves presented herein permit evaluation of the ratio of aircraft thrust minus drag to ideal thrust in order to

facilitate the choice of inlet for its propulsion system. An equation for the ratio of incremental changes in inlet pressure recovery to associated incremental changes in configuration drag coefficient required to maintain a constant level of thrust minus drag is presented. This equation is also applicable to the determination of inlet mass-flow ratio for maximum thrust minus drag. The method applies to any air-breathing engine where total-pressure ratio, total-temperature ratio, and air-flow requirements are known.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, June 29, 1954

APPENDIX A

DERIVATION OF THRUST PARAMETER FOR AIR-BREATHING PROPULSION SYSTEM

For a power-plant installation with a completely expanded exhaust nozzle at each flight condition (fig. 1), the internal thrust is given by

$$F = m_e v_e - m v_0 \tag{A1}$$

In coefficient form based on the free-stream tube area of the air flow passing through the engine, equation (A1) becomes

$$C_F = 2\mu \frac{v_e}{v_0} - 2$$

or

$$C_F = 2\mu \frac{M_e}{M_0}\left[\frac{\gamma_e R_e T_e}{\gamma_0 R_0 T_0}\left(\frac{1 + \frac{\gamma_0 - 1}{2} M_0^2}{1 + \frac{\gamma_e - 1}{2} M_e^2}\right)\right]^{1/2} - 2 \tag{A2}$$

Transposing terms and converting exit Mach number to pressure-ratio relations yields

$$\frac{C_F + 2}{\mu\left(\frac{\gamma_e R_e T_e}{\gamma_0 R_0 T_0}\right)^{1/2}} = \frac{2\left(1 + \frac{\gamma_0 - 1}{2} M_0^2\right)^{1/2}}{M_0}\left\{\frac{2}{\gamma_e - 1}\left[1 - \left(\frac{p_e}{P_e}\right)^{\frac{\gamma_e - 1}{\gamma_e}}\right]\right\}^{1/2} \tag{A3}$$

For a completely expanded exhaust nozzle, $p_e = p_0$ and, for equation (A3):

$$\frac{p_e}{P_e} = \frac{p_0}{P_0}\frac{P_0}{P_e}$$

Consequently, equation (A3) can be written as

$$\frac{C_F + 2}{\mu\left(\frac{\gamma_e R_e T_e}{\gamma_0 R_0 T_0}\right)^{1/2}} = \frac{2\left(1 + \frac{\gamma_0 - 1}{2} M_0^2\right)^{1/2}}{M_0}\left\{\frac{2}{\gamma_e - 1}\left[1 - \left(\frac{p_0}{P_0}\frac{P_0}{P_e}\right)^{\frac{\gamma_e - 1}{\gamma_e}}\right]\right\}^{1/2} \tag{A3a}$$

and the thrust parameter $\dfrac{C_F + 2}{\mu\left(\dfrac{\gamma_e R_e T_e}{\gamma_0 R_0 T_0}\right)^{1/2}}$ becomes a unique function of free-stream Mach number and over-all power-plant total-pressure ratio, P_e/P_0. For this analysis, it was assumed that $\mu\left(\dfrac{\gamma_e R_e}{\gamma_0 R_0}\right)^{1/2} = 1$, $\gamma_e = 1.32$, and $\gamma_0 = 1.40$.

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The over-all total-pressure ratio P_e/P_0 may be expressed as

$$\frac{P_e}{P_0} = \frac{P_e}{P_2}\frac{P_2}{P_0} \tag{A4}$$

where P_2/P_0 is the inlet pressure recovery and P_e/P_2 is the engine pressure ratio. Thus for a given engine pressure ratio, the effect of inlet pressure recovery on over-all total-pressure ratio, and, consequently, thrust parameter, can be determined.

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APPENDIX B

EVALUATION OF VARIATION OF INTERNAL THRUST WITH INLET PRESSURE RECOVERY

If, for a given engine at a given operating point, the engine total-pressure ratio P_e/P_2, total-temperature ratio T_e/T_2 $(T_e/T_2 = T_e/T_0)$, and free-stream Mach number M_0 are known, the variation of F/F_i with P_2/P_0 can be determined from equation (A3a) with the aid of equation (A4). For $P_2/P_0 = 1.0$, $\frac{P_e}{P_0} = \frac{P_e}{P_2}$. When these values of P_e/P_0, T_e/T_2, M_0, and p_0/P_0 (function of M_0) are inserted into equation (A3a), a value of $C_{F,i}$ is obtained. For values of $P_2/P_0 < 1.0$, C_F can be determined by inserting the appropriate values of P_e/P_0 into equation (A3a) as determined from equation (A4). For each value of P_2/P_0 assumed then a value of $C_F/C_{F,i}$ can be obtained. Now

$$\frac{F}{F_i} = \frac{C_F q_0 A_0}{C_{F,i} q_0 A_{0,i}} = \frac{C_F A_0}{C_{F,i} A_{0,i}} \tag{B1}$$

and

$$\frac{A_0}{A_{0,i}} = \frac{m_2}{m_{2,i}} = \frac{P_2 A_2 (A^*/A_2)}{P_{2,i} A_2 (A^*/A_2)_i}$$

At a given engine operating point, M_2 is constant; consequently, $A^*/A_2 = (A^*/A_2)_i$ and, $A_0/A_{0,i} = P_2/P_{2,i} = P_2/P_0$. Therefore,

$$\frac{F}{F_i} = \frac{P_2}{P_0}\frac{C_F}{C_{F,i}} \tag{B2}$$

Variation of F/F_i with P_2/P_0 is shown in figure 2 (solid line) for $M_0 = 2.0$, $P_e/P_2 = 2.38$, and $T_e/T_2 = 4.0$. The curve is seen to be nearly linear and could be approximated with a straight line (long dashes) drawn between the end points at $F/F_i = 0$ and 1.0. This approximation would introduce some error, however, particularly in the middle region of the curve. A more accurate approximation can be obtained by assuming a linear variation of F/F_i for values of P_2/P_0 between 0.667 and 1.0 (short dashes) which is in good agreement with the actual

curve (solid line) for most of the useful range of inlet pressure recoveries obtainable at free-stream Mach numbers up to about 3.0. This straight-line approximation for F/F_i can then be written

$$\frac{F}{F_i} = k \frac{P_2}{P_0} + (1 - k) \tag{B3}$$

where k is the slope of the curve and is given by

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$$k = \frac{1 - (F/F_i)_{P_2/P_0=0.667}}{0.333} = 3 - 2\left(\frac{C_F}{C_{F,i}}\right)_{P_2/P_0=0.667} \tag{B4}$$

For convenience, a set of curves obtained from equation (B4) and the equations of appendix A have been presented in figure 3 for the determination of k.

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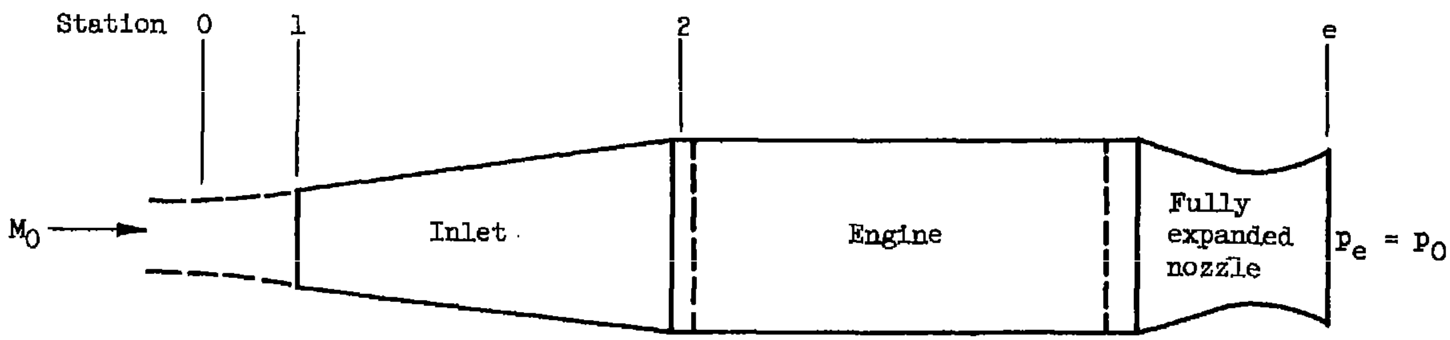

Figure 1. - Schematic diagram of propulsion system.

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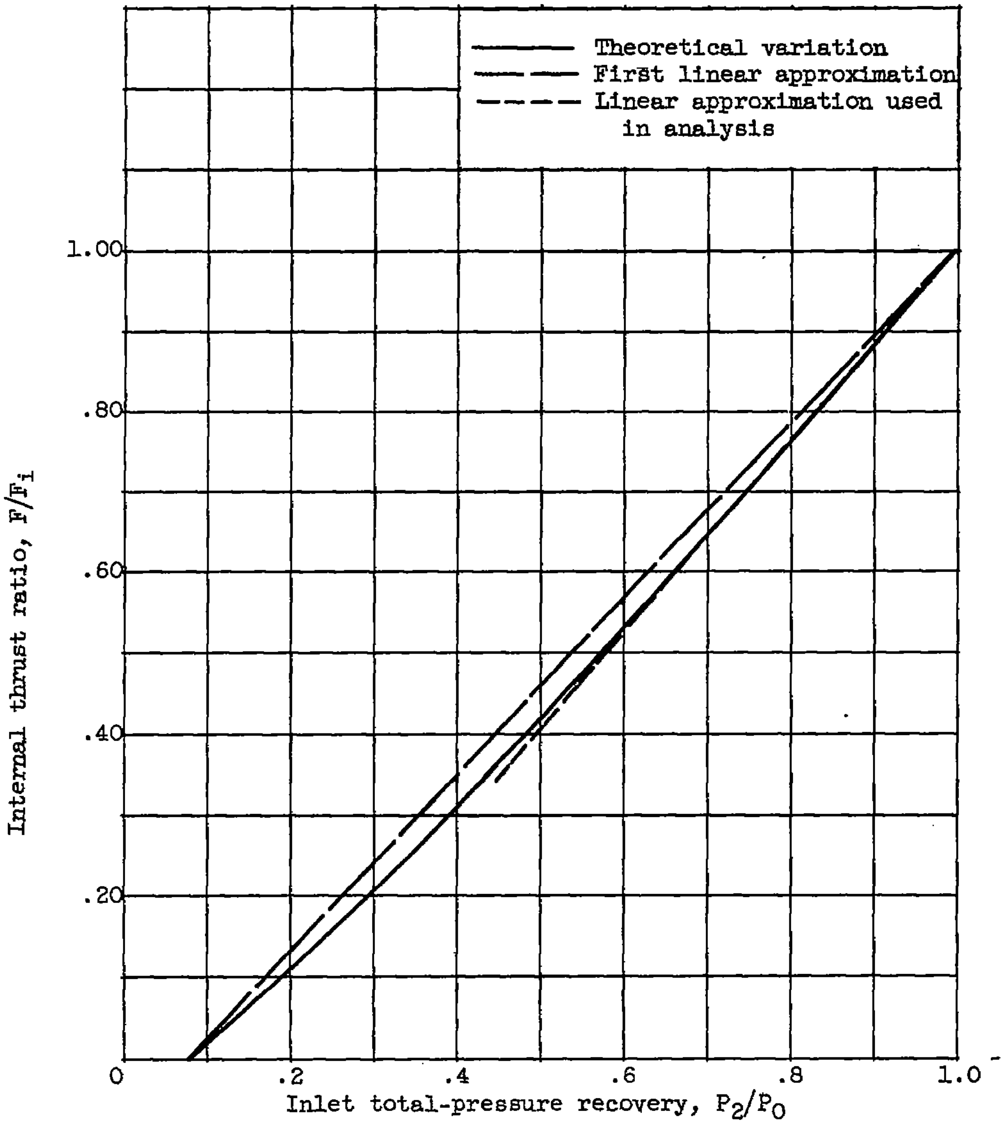

Figure 2. - Thrust ratio variation. Engine total-pressure ratio, 2.38; engine total-temperature ratio, 4.0; free-stream Mach number, 2.0.

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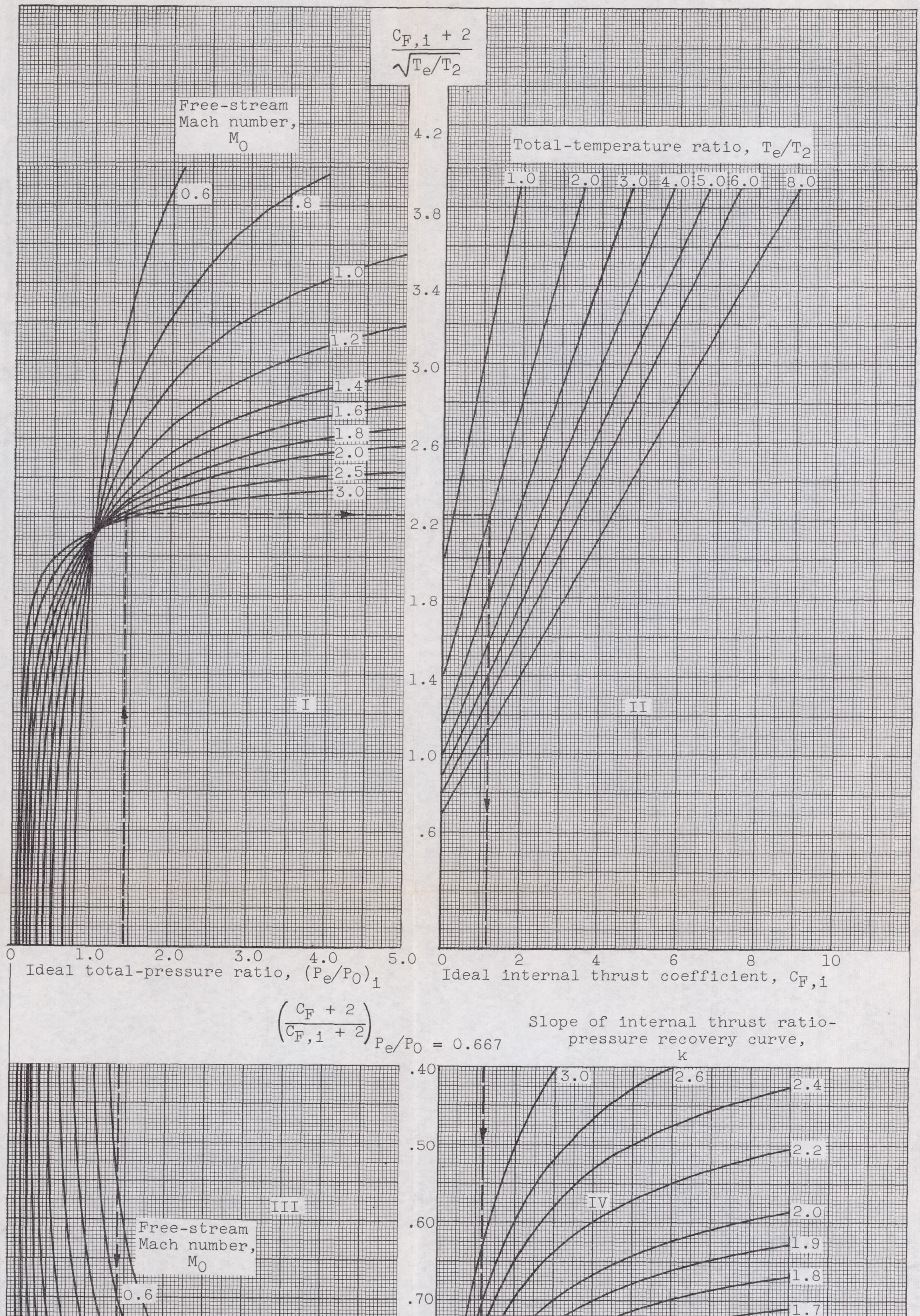

Figure 3. - Thrust-parameter functions.

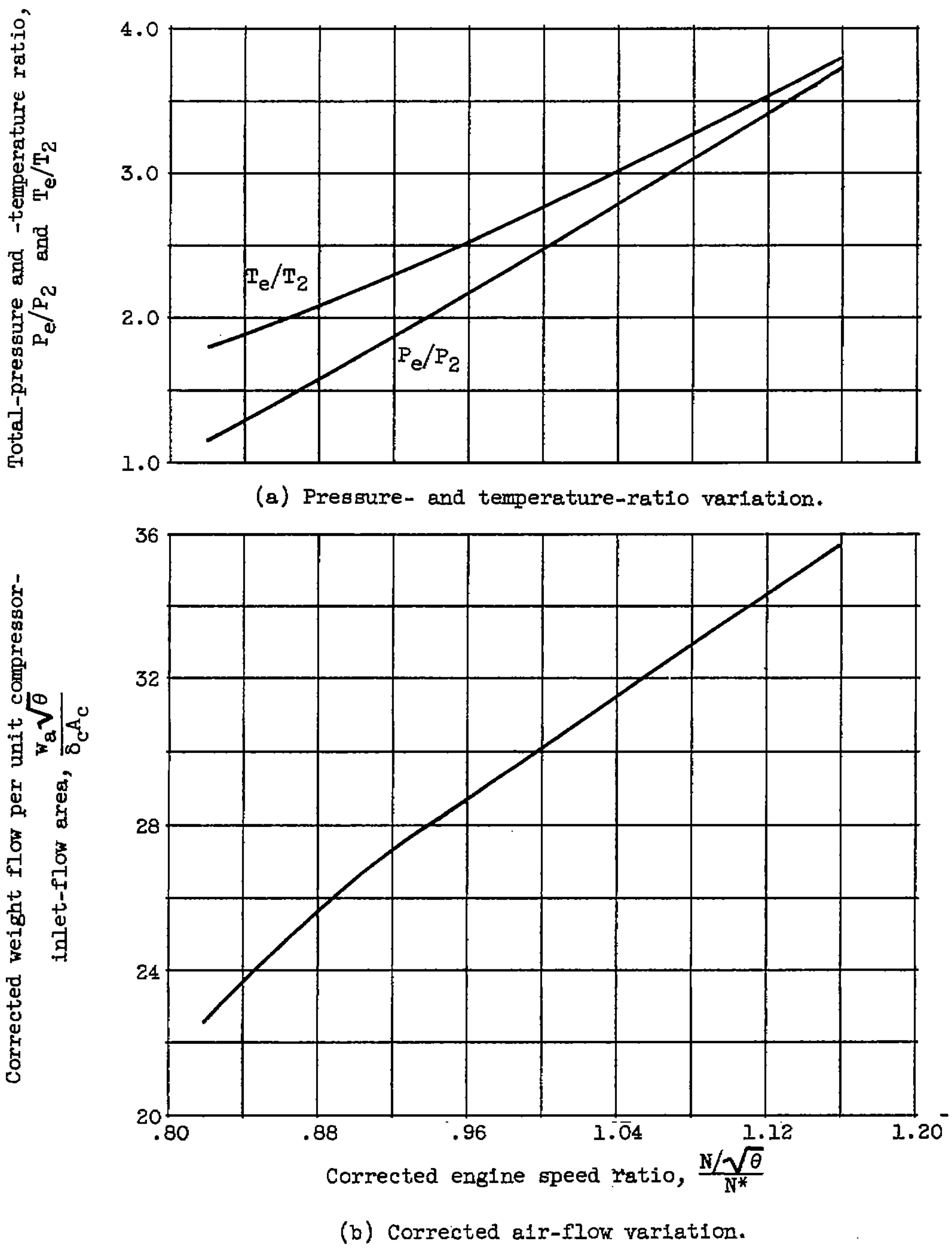

Figure 4. - Generalized engine characteristics for typical turbojet engine.

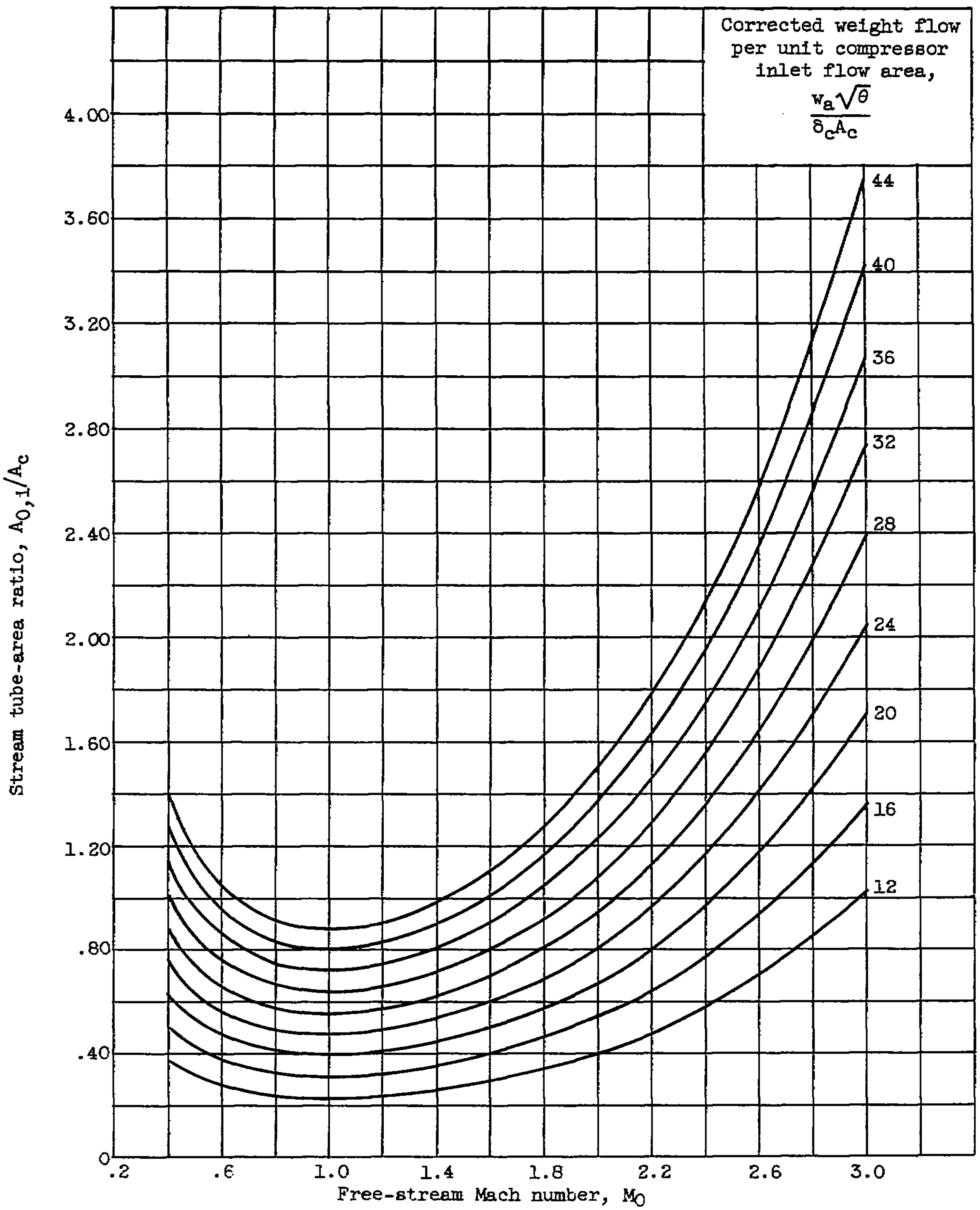

Figure 5. - Stream tube-area ratio for ideal inlet operation.

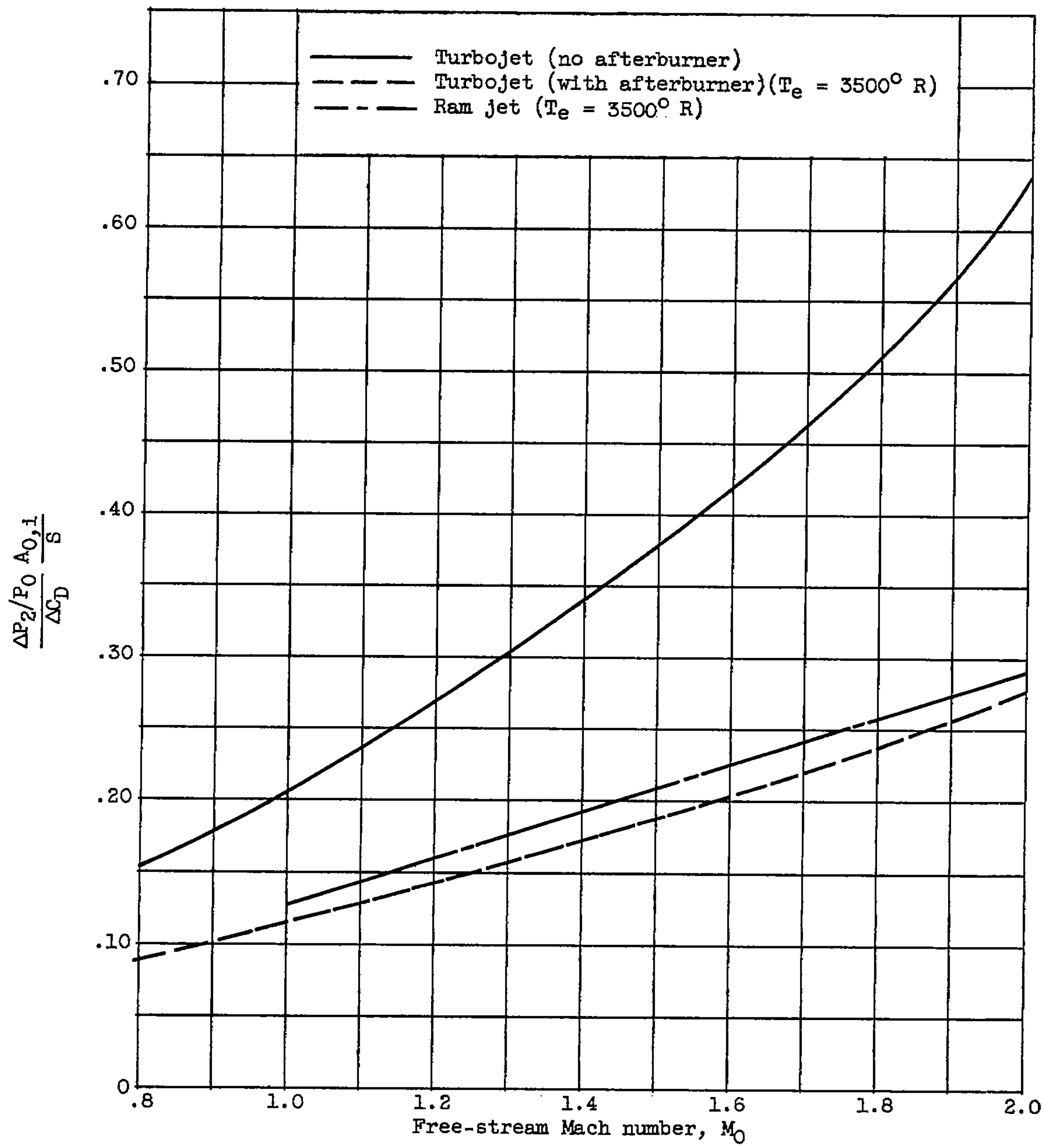

Figure 6. - Parameter relating ratio of incremental pressure recovery to incremental drag coefficient to maintain constant thrust minus drag. Isothermal region of atmosphere.

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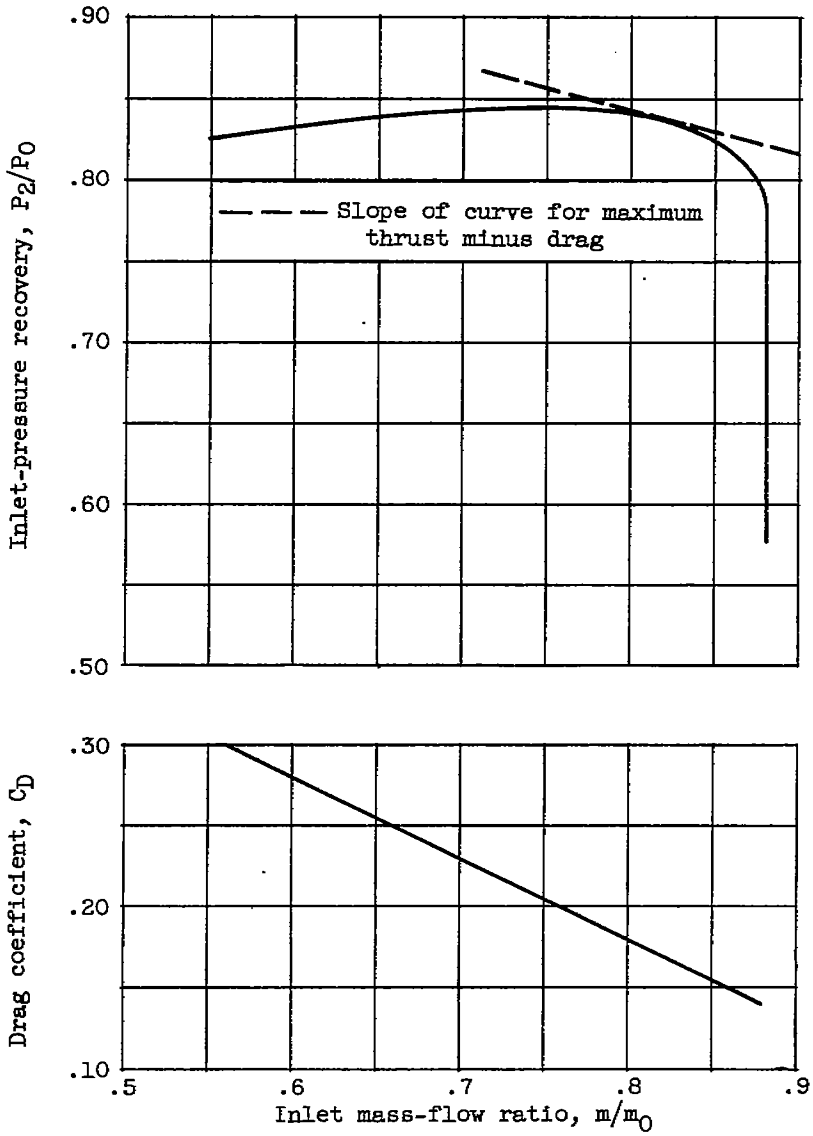

Figure 7. - Typical inlet-pressure recovery and drag coefficient variation at free-stream Mach number 2.0. Drag coefficient based on inlet capture area.

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